DEVELOPMENT AND DEMONSTRATION OF A 600 SECOND MISSION AVERAGE ARCJET

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Abstract

Recent work conducted at Rocket Research Company (RRC) under the NASA sponsored Arcjet Thruster Development program (NAS3-26055) has resulted in a 550 hour demonstration of a hydrazine arcjet at a mission average special impulse (Isp) of greater than 600 s. The laboratory type arcjet thruster was operated at 1800 W through a flow rate schedule consistent with a flight application. The thruster demonstrated stable, high efficiency operation for the duration of the test. The development effort leading to the demonstration addressed the operational and life issues associated with achieving high performance. As a result of the development activity, a wide range of performance was also achieved. Performance levels of greater than 575 s at 1000 W and greater that 675 s at 2000 W were demonstrated. This paper discusses the problems encountered with extending low-power arcjet performance. Increased thruster component temperatures caused new life issues to be identified, along with low propellant flow rate stability limits. The elevated temperatures caused significant changes in the electrode geometry for a thruster based on the state-of-the-art tungsten anode designs. Three paths to solving the problem were attempted, including lowering the operating temperature through improved heat rejection, mechanical design changes to reduce thermal stresses, and higher strength materials selection. The success of this program provides substantial evidence that an arcjet capable of greater than 600 s mission average specific impulse will be available in the near term for flight qualification and application.

Introduction

In 1983, Rocket Research Company under an internal effort investigated the concept of using low-power arcjet thrusters for north-south stationkeeping (NSSK) on geosynchronous communication satellites. The initial results showed that by reducing the NSSK propellant mass, the user was given three cost saving options. The first consisted of using a smaller launch vehicle. In the other two options the size of the spacecraft remained the same but either operating life could be increased through more efficient use of propellant or some of the propellant could be offloaded to make room for electronics. In 1984, NASA began supporting both an in-house and a contracted effort to develop a low-power arcjet system. The NASA program was system oriented and addressed not only the thruster, but also the power processor and propellant system.

In 1984, RRC was selected as the prime contractor for NASA's Arcjet Thruster Research and Technology (ATRT) Program. The goal of that program was to develop an arcjet system with a specific impulse of 450 s at a power level of 1.4 kW. The main role of the contracted effort was to develop a flight qualifiable design, while the in-house portion of the program concentrated on solving performance and life issues. One of the earliest problems encountered in the program was the development of a non-erosive starting technique. That problem which plagued earlier programs was solved through a novel method of power processing(1,2). By 1987 thruster life issues at the program performance target of 450 s had been solved and a 1000 h/500 cycle endurance test was completed on laboratory model equipment(3). In 1989, RRC completed an 811 h/811 cycle performance test using engineering design model (EDM) components, including the arcjet thruster, power processor, and

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gas generator. This effort resulted in a highly successful transfer of technology to commercial use. Using the NASA/RRC developed EDM system as a baseline, RRC and GE/Astrospace Division (now Martin Marietta Astro Space – MMAS) developed a 1.8 kW, 502 s minimum nominal mission average (NMA) specific impulse, flight-qualified system which has been baselined on several series MMAS Series 7000 communications spacecraft. NASA has continued to contribute to the arcjet flight system by providing spacecraft integration support. The large vacuum facilities at NASA Lewis Research Center have been used to measure the radiated electromagnetic emissions, to examine thruster/spacecraft surface interactions and to investigate spacecraft charging/discharging phenomena. Measurements of the electron number density and temperature of the thruster plume have been used to formulate models to examine the effects on the communication signal.

In 1991, NASA began the Arcjet Technology Development (ATD) Program. The performance target of the current program is 600 s NMA at 1.8 kW into the thruster for 1000 h. With funding from the NASA ATD program, RRC has addressed the issues necessary to reach the goal.

The paper details the effort to identify and resolve the above issues and also documents the results of the performance and life demonstration. The thruster hardware, test facilities test procedures and data measurement methods are also described. The effort culminated with a successful 550 h demonstration test of a 600 s NMA Isp arcjet. The thruster operated with hydrazine propellant at 1800 W and demonstrated stable operation throughout the flow range.

Apparatus and Procedures

An extensive test effort was conducted to investigate the effects of the proposed methods to resolve the constrictor closure issue. The effort used a laboratory type thruster test-bed that allowed for rapid change out of modified nozzles. The following section is a description of the hardware, facilities and test procedures.

Arcjet Assembly

The test apparatus included an arcjet assembly (Fig. 1) and a RRC owned power conditioning unit (PCU). The primary elements of the arcjet assembly included the arcjet thruster, hydrazine gas generator, propellant valve and fluid resistor. The arcjet thruster used for this effort was a laboratory type, radiation cooled design developed specifically for use on this program. The baseline thruster was modelled after the SOA arcjet design with a tungsten anode. The design incorporated characteristics that were very similar to flight type thrusters. The internal features such as electrode configuration and isolation, flow passages and vortex injection techniques were all based on previously successful designs. The thruster thermal design also simulated the flight type design. The purpose of this approach was to take advantage of the best attributes of the existing flight design and to also minimize the effects of the eventual transition from a laboratory model to a flight thruster.

The modular features of the thruster provided easy and reliable exchange of various thruster components. The anode could be removed from the thruster by unscrewing the large nut that holds the anode to the body. With the anode removed, all of the internal components could also be removed and changed to allow parametric studies of the thermal effects.

The valve, gas generator and fluid resistor were identical components to those used on the previous NASA arcjet. The gas generator contained a catalyst chamber which decomposed liquid hydrazine into nitrogen, hydrogen, and ammonia gases. The fluid resistor was a passive device which established the system flow rate based on the upstream feed pressure.

Test Facility and Instrumentation

Testing of the arcjet was conducted in Chamber 11 of the RRC Electric Propulsion Test Facility. The chamber is 2.44 m in diameter and 2.44 m long, constructed of steel, and has a water cooling jacket. The vacuum pumping system included a mechanical pump with a capacity of 13,400 cfm. Over the propellant flow rates tested (25 to 45 mg/s), the chamber pressure ranged between 20 to 50 mTorr. Thrust was measured on a swing arm thrust stand. The thrust stand was operated in a null displacement mode using a combination LVDT/linear actuator measurement system. Error due to hysteresis effects was minimized by maintaining nearly zero displacement of the thrust arm.

The thruster was mounted on a heat exchanger plate which was fixed to the thrust arm. This allowed a
simulation of the mounting interface expected for a flight application. Lines for electrical power, hydrazine, conditioning fluid, and instrumentation were integrated into torsional flexures which were aligned with the thrust arm axis of rotation.

Arcjet voltage and current measurements were made from instrumentation designed to interface with a power cable which was modified for testing. These modifications allowed the cable to be assembled with a cabinet housing, two current transformers, each with different frequency response characteristics, and a broadband voltage divider circuit to allow steady-state and transient measurements to be made.

The propellant delivery system simulated spacecraft requirements. The propellant tank was pressurized with high purity helium. Flow rate was measured with a coriolis effect mass flowmeter and a remotely controlled sightglass system was used less frequently for redundant measurements during performance mapping. A DC power supply rated at 150 V and 70 A was used to supply the PCU input power.

Additional instrumentation included strain gage pressure transducers, chromel-alumel thermocouples, and a digital storage oscilloscope for recording high frequency voltage and current measurements. Test control and data acquisition were performed by a micro-computer based system integrated with a 16 channel digitizing data logger. Software was developed to allow complete control of the arcjet system functions via the computer.

Experimental Procedures

The program's test activity was broken into two groups. The first, concentrated on short term investigations such as performance mapping and observing the effects on performance of thermal design modifications, electrode geometry changes and variations in input power. The second group addressed long term and multi-cycle issues such as extended high temperature exposure of thruster components or the effects of thermal and electrical cycling of the electrodes.

Short Term Investigations

The objective of the short term investigations was to measure the effect of electrode, thermal design or input parameters changes on the performance or operational characteristics of the arcjet. The modular arcjet allowed quick changes to the thruster configuration to be tested.

The arcjet was fully instrumented and placed in the test set up as described above. A complete system checkout was made to verify thruster and data acquisition and control system integrity. The vacuum level was maintained from 10 to 50 mTorr depending upon the propellant flow rate during the test. Unpowered runs were conducted at the beginning of the test for each new configuration to establish a baseline set of parameters that indicated the condition of the thruster. Unpowered tests were conducted periodically during and at the end of the test series as non-intrusive health checks of the thruster.

Performance mapping was accomplished by operating the thruster for one hour to assure thermal equilibrium. At the end of the run, the data was averaged and corrected for any thermal related zero-shifts to assure accuracy. Tests were conducted at three to four discrete flow rates for each power level to establish performance trends. Data accumulated for the input power comparisons was taken at 1000, 1250, 1500, 1800 and 2000 W. The thermal effects tests were all conducted at 1800 W input power.

Long Term Investigations

The long term investigations simulated thruster operation expected during a flight application. The thruster was operated for multiple one hour cycles. One half hour spans between runs provided sufficient cool down periods. The experimental procedures for these investigations were similar to those for the short term effort. The thruster was placed in the same set up with similar instrumentation.

The key difference was automation. The extended duration tests were controlled by a computer system. The computer controlled propellant flow and power to the thruster. Data was recorded every five (5) minutes. At the end of each cycle, the computer collected an end of run (EOR) data sample. The computer also monitored the health status of the test system. Limits were set for all of the critical parameters such as maximum power level, minimum flow, minimum arc voltage and high temperature limitations of critical components. If any of the limits were exceeded, the system would automatically record an EOR data sample and safely shut
the test down. This capability allowed the extended duration tests to be operated automatically 24 hours a day.

**Demonstrated Performance**

The goal of the NASA Arcjet Thruster Development program is to expand the performance and life capability of low power arcjet technology. A specific program objective was to investigate arcjet operational stability at low propellant flow through a wide power range. The investigation resulted in an early demonstration of 550 s mission average Isp at 1000 W and 600 s mission average specific impulse at 1800 to 2000 W. Figures 2 and 3 illustrate the performance capability of the improved arcjet technology developed in this program. Peak performance levels of 575 s Isp at 30% efficiency at 1000 watts and 675 s at 30% efficiency and at 2000 watts were demonstrated.

Figures 1 and 2 also illustrate the extension of the performance and operating power range over the previous state of the art technologies(4,5). The performance levels demonstrated at 1800 to 2000 W represent improvements of approximately 100 s. The stable high performance demonstrated at 1000 W represents an improvement of approximately 150 s.

**Life Limiting Issues**

An important part of the program’s goal was to demonstrate the high performance levels for durations typical of potential flight requirements. Along with mapping thruster performance, in-situ and post test observations were made in search of potential life limiting phenomenon. Four major life issues were identified.

The first is convergent anode surface erosion caused by low flow starts. As the flow rate is decreased to accommodate higher specific power levels and higher performance, starting the arcjet without significant erosion becomes increasingly difficult. Observations made during the performance verification tests indicated that significant anode erosion occurred at starting flow rates below 34 mg/s.

The other three issues: insulator degradation, cathode degradation and constrictor closure are all thermally related. The thermal loading of the high performance arcjet anode is considered to be the main limiting factor for arcjet life. Surface temperatures at normal operating conditions for the thruster are approximately 500°C greater than the standard flight type thruster. Thermal model predictions indicated that internal temperatures near the constrictor could be as high as 1925°C and insulator temperatures near the cathode could be similar.

**Low Flow Startup Erosion**

The start process is characterized by the following steps.

1. Arc initiation – Paschen breakdown occurs at steady state flow rate. Breakdown voltage may be 2500 to 4500 volts. Arc voltage drops quickly to 30 to 50 volts as current is established and arc spot attachment moves to the edge of the constrictor.

2. Arc transition – Gas flow forces the arc through the constrictor. The arc voltage increases quickly to a range of 75 to 125 volts depending upon the flow rate and power level.

3. Arc Stabilization – Gas expansion in the nozzle allows diffuse arc attachment. Arc voltage remains in the 75 to 125 volt range.

Extended operation in step #1, where the arc voltage remains at approximately 35 to 50 volts, is commonly referred to as "low mode". The amount of convergent anode surface degradation is proportional to the amount of highly concentrated thermal energy deposited into the surface:

\[ V_{arc} \cdot I_{arc} \cdot T_{lm} = \text{deposited arc energy} \]

where

\[ V_{arc} = \text{Arc voltage (volts)} \]

\[ I_{arc} = \text{Arc current (A)} \]

\[ T_{lm} = \text{Low mode duration (s)} \]

For the thruster/PCU configuration used in these tests, low mode durations of approximately 50 ms or less typically resulted in clean starts with minimal erosion. As low mode duration increased, the amount of energy concentrated on the converging anode surface also increased, raising the potential for tungsten degradation and possible erosion of material.

High current starts have a similar erosive effect as extended low mode starts. The initial performance tests used a power conditioning unit (PCU) that
delivered 12 A of current during the start process. RRC reduced low flow start erosion by adjusting the steady state converter of the PCU to deliver 3 to 6 A of current. A start current level of 6 A provided consistent low erosion starts at flow rates above 32 mg/s. Post test visual inspections of the anodes indicated improved erosion characteristics.

Additional methods to improve start erosion are currently being investigated\(^\text{(13)}\). Because those methods are not yet perfected, the minimum start flow rates maintained for the performance and extended duration tests was 38 mg/s. This flow rate corresponds to the minimum required for the 2000 watt, 600 s mission average Isp as shown in Table 1.

**Table 1. Start Flow Rate Requirements**

<table>
<thead>
<tr>
<th>Mission Average Specific Impulse (s)</th>
<th>Power Level (W)</th>
<th>Min. Flow Rate Required for Start (mg/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>525</td>
<td>1000</td>
<td>26.5</td>
</tr>
<tr>
<td>545</td>
<td>1200</td>
<td>29.3</td>
</tr>
<tr>
<td>575</td>
<td>1400</td>
<td>32.5</td>
</tr>
<tr>
<td>600</td>
<td>1800</td>
<td>34.0</td>
</tr>
<tr>
<td>600</td>
<td>2000</td>
<td>38.0</td>
</tr>
</tbody>
</table>

Table 1 lists the initial flow requirements to achieve a desired peak performance level at different power levels. The initial flow requirement is higher than the steady state flow rate by approximately 5 to 10% because of the change in pressure drop characteristics through the constrictor before and after arc initiation. Based on a 2000 W input case, the minimum initial flow requirement to achieve 625 s Isp peak performance (600 s mission average) is approximately 38 mg/s.

**Thermal Related Life Issues**

**Insulator Degradation**

Final inspection of thruster components after each of the above tests revealed significant degradation of the internal cathode insulators. The alumina material showed visual signs of severe heat loading on the forward face. The insulators were found fused to their respective cathodes upon disassembly. Thermal cycling of the fused components resulted in the failures of the insulators.

RRC resolved the insulator problem by modifying the insulator design to provide minimal contact between the high temperature regions of the cathode and the insulator. Essential contact between the two components for the purpose of centering the cathode, was maintained in a cooler region for both the cathode and the insulator.

**Cathode Degradation**

An additional life issue observed after the above tests was the high rate of cathode tip recession. Figure 4 illustrates the cross-sectional tip profile at various accumulated hours of cyclical operation. The cathode as operated at 1800 W and at performance levels above 580 s Isp. Figure 5 compares the tip recession rate with that of SOA cathodes operated at 1600 W and at performance levels below 520 s.

The initial rate of recession for the high performance cathodes is approximately a factor of three (3) greater than the lower performance cathodes. The recession rate decreases, however over time due to the exposure of a larger cathode tip surface area. The rate of recession after approximately 200 h has been reduced to less than a factor of two greater than the low performance rate. Extrapolating to 1000 h, the expected tip recession for a high performance arcjet operating at 1800 W is 0.90 mm (0.034 inches). This represents a tip recession approximately 0.25 mm (0.010 inches) greater than observed during previous tests. Despite the large potential arc gap, no problems such as breakdown failures or arc instabilities were observed.

**Constrictor Closure Phenomenon**

The final life limiting issue uncovered during the tests was the continual reduction of constrictor area during the life of the thruster. Examination of the arcjet operational parameters during the high performance investigations revealed a shift over time in the chamber pressure versus flow characteristics. This characteristic is nearly linear and very repeatable for a thruster operating at a constant power level and an unchanging electrode configuration. The approximate value of constrictor area, therefore, can be monitored by observing chamber pressure.

A more representative term of constrictor area observed during thruster operation is the parameter
of \( P_C / \sqrt{m} \). This term and what it represents is as follows:

Thrust chamber pressure in a one-dimensional isentropic flow with heat addition upstream of the throat, is a function of mass flow rate, constrictor area and arc power. The relationship is represented by the following equation:

\[
P_C = \frac{1}{(A \cdot \gamma)} \cdot \sqrt{\frac{\dot{m} P}{(\gamma + 1/2)}}
\]

where:
- \( P_C \) = thrust chamber pressure
- \( A \) = constrictor area
- \( \gamma \) = ratio of specific heat
- \( \dot{m} \) = mass flow
- \( P \) = power input

At constant power and assuming a constant ratio of specific heat, the equation can be written as:

\[
\frac{P_C}{\sqrt{\dot{m}}} \propto A \cdot \sqrt{m}
\]

or:

\[
\frac{1}{\alpha} = \frac{P_C}{A \cdot \sqrt{m}}
\]

Therefore, a graph of \( P_C / \sqrt{\dot{m}} \) versus time is an inverse representation of the constrictor area. The ratio of the parameter during test with its initial value is an approximation of the constrictor area ratio:

\[
A / A_o = \frac{P_{co}}{P_C} \cdot \frac{\sqrt{m}}{\sqrt{m_o}}
\]

It is considered only an approximation because the measured chamber pressure is not the true chamber pressure as identified in the above equations. The indicated \( P_C \) is observed at the inlet to the thruster and does not account for the pressure losses across the thruster inlet, internal passages and the vortex injectors.

From the area ratio and the measurement of the initial constrictor diameter, an approximation of the constrictor diameter during test can be made.

To provide a standard method of comparing multiple thrusters, the real time data is normalized to the initial data point. Figure 6 is a graph of normalized \( P_C / \sqrt{m} \) for the SOA thruster configuration tested at 600 s. The graph illustrates the dramatic constrictor closure rate experienced as compared to a maximum acceptable closure rate. At the end of less than 50 hours, the constrictor area was reduced by greater than 40%. In a flight application where the thruster feed pressure decreases slowly with time, this change in area could result in a self damaging process. A reduced diameter would restrict the flow further causing a greater potential for damage. The importance of resolving this life issue, dominated the program effort.

**Constrictor Closure Investigation**

**Fundamental Process Identification**

Reduction of the constrictor area can only occur if there is movement of material into the constrictor region. Three possible mechanisms were considered: swelling of reduced density tungsten near the constrictor walls, the build-up of material transferred from an erosive process upstream of the constrictor such as the cathode or the convergent section of the anode and finally, a permanent movement of solid tungsten material into the constrictor region.

The first process was eliminated by chemical and metallurgical analysis. The results of the chemical analysis indicated the material near the constrictor was pure tungsten of a high density. No voids were observed in the metallurgical analysis to indicate a swelling or degradation of the material.

To investigate the material transfer theory, RRC conducted a Scanning Electron Microscope (SEM) and Energy Dispersive Spectroscopy (EDS) analysis of the internal surface of an anode that was tested for 50 hours. The anode experienced a constrictor diameter reduction of 0.051 mm (0.002 inches). Inspections revealed surface erosion occurring at the leading edge of the constrictor. Deposition of tungsten material was evident in the downstream edge of the constrictor and in the nozzle region. The thickness of the deposited material was estimated to be approximately 0.0051 mm (0.00002 inches) in the nozzle region and less in the constrictor area. This represented a deposition layer that was 2 orders of magnitude less than the observed diameter reduction. It was concluded that tungsten deposition was occurring, but at a rate that did not account for the
constrictor closure observed and was therefore considered a secondary contributor.

A complete thermal and structural analysis of the anode was conducted to address the closure phenomenon. Axi-symmetric thermal and finite element structural models were developed of the solid tungsten anode. Results of the analysis suggested that extremely high stress levels resulting from high thermal gradients exist near the constrictor surface. These high stress levels combined with elevated material temperatures result in an inward yielding of the material.

The amount of material deformation predicted by this method was smaller than observed during test, however. The analysis also showed that the yielding process would decrease after a few cycles. A first order creep model was therefore investigated in an attempt to further understand the closure process. Review of the analytical and experimental data resulted in the following assessment:

The closure process can be broken into three distinct elements corresponding to the arcjet operational cycle:

**Thermal Transient:** The period of time between startup and steady state. The arcjet’s transient behavior may contribute the majority of the constrictor deformation due to the rapid increase in constrictor temperature and the high thermal gradients.

**Steady State:** At steady state, the arcjet is at thermal equilibrium under operational conditions. Elastic strains that are present during the steady state may become inelastic over time resulting in a contribution to the closure.

**Cool Down:** The period of time where the thruster is off and returning to room temperature equilibrium. The cool down cycle may be more important than anticipated because of its potential to reverse some of the closure.

The problem is very complex. Analysis to accurately predict the closure rates and specific deformations requires accurate knowledge of the thermal gradients which are not available. The analysis, therefore, was used as a qualitative tool to study effects. In summary, the closure process is believed to be the result of high hoop stresses induced by the radial thermal gradients. The stresses combined with the elevated material temperatures and subsequent reductions in strength, result in both a short term yielding and long term creep phenomenon. The cumulative result is an inward movement of material.

**Proposed Solutions**

Three potential strategies were proposed to resolve the closure issue:

1. Decrease the overall operating temperature of the anode, thereby increasing the strength of material beyond the area of concern.
2. Relieve the stress by an improved structural design.
3. Replace the pure tungsten material with a material that has relatively the same thermal properties, but has significantly increased strength.

Under the ATD program, over 20 anode configurations were tested in an attempt to understand and resolve the constrictor closure phenomenon. The following test configurations shown in Fig. 7 and highlights of the complete effort are described below.

The results use data from the baseline high performance thruster to determine the progress of the closure investigation. New data is also compared to the maximum acceptable closure rate determined from previous efforts.

**Method 1 – Thermal Approach**

The purpose of this method was to reduce anode temperatures and maintain high strength material properties as a result. The anode may be cooled to lower temperatures by passive or active heat rejection. Active heat rejection such as regenerative cooling was not considered in this effort.

Heat can be removed passively from the anode area by either better conduction away from the anode or by increased anode radiative capability. Increasing the conduction capability was not considered in the initial analysis due to its direct impact on other portions of the thruster.

Two alternatives exist for improving the radiative capability. The first is to increase the surface emissivity. The second is simply by increasing the radiative area.
A thermal analysis of the solid tungsten anode investigated the effect of changing the anode surface emissivity. The thermal model analyzed an anode with a surface emissivity of 0.5 and also with 0.999 for demonstration purposes. Higher emissivity decreased the anode temperatures by approximately 400 degrees. The actual temperature reduction will be less with a functional anode coating has an emissivity of approximately 0.75. A reduction in material temperatures will increase the strength of the anode material.

Increasing the radiative area is not as desirable as increasing the emissivity because of the resulting increase in weight. The thermal effect would be similar, but, the added weight could adversely impact the structural/vibrational integrity of the thruster.

A quick method of increasing the heat rejection on test hardware was accomplished with the high emission sleeves shown (Fig. 7). A silicon carbide sleeve can be lightly pressed onto the anode to provide an increase in emissivity (SiC emissivity = 0.9). Tests conducted in the previous NASA arcjet program showed that with the use of the sleeve, exterior anode temperatures can be reduced by approximately 100°F. Adding of graphite disks also provided passive heat rejection by increasing radiating area as well as the surface emissivity.

Tests were conducted on a thruster anode with and without additional heat rejection to determine the potential gains of cooling the anode. The anode was shown to be sensitive to the temperature. The rate of constrictor closure, as indicated by the normalized $P_c / \sqrt{\dot{m}}$, changed with the operating condition. The rate of closure increased with increasing performance and the resulting increased anode temperatures. To investigate the result of artificially manipulating temperatures, the anode was equipped with a graphite radiating disk (Fig. 7). The radiator reduced the exterior anode temperatures by an average of 300°C. The test results indicated a reduction in the rate of constrictor closure. The performance of the anode, however, also dropped by 15 to 20 s Isp.

Observations of the closure rate at the same performance level, however, indicated an improvement. The rate of closure was less for the radiatively cooled design. For example, the closure rate for the radiatively cooled anode operating at 600-s Isp was approximately 60% of the closure rate for the non-cooled anode. The cooling of the anode does therefore provide an improvement, but did not fully resolve the closure process and had the negative impact of reduced thruster efficiencies.

**Method 2 – Improved Structural Design**

Another possible method of relieving the stress in the anode is by altering the structural design to allow further expansion of the inner material. A structural analysis was conducted of a thin wall design (Fig. 7). The results of the analysis show that the magnitude of the stress decreases with decreasing wall thickness. Actual values of stress reduction depend on the amount of thinning and temperatures.

An unfavorable aspect of this approach was the very high anode temperatures resulting from the decreased radiating area. The high temperatures and reduced material strength could offset the gains made by thinning the anode walls. Anode designs combining Methods 1 and 2, however, may provide the structural flexibility required for relief of the internal stresses, and the radiative capability to maintain lower temperatures. Results of a test using this approach are discussed in the test section.

The above represent external approaches to reducing the stress level in an anode. An approach to modifying the internal geometry of an anode was also investigated. Results of a test on an anode with internal relief slots are described in the test section.

RRC investigated two basic approaches to providing the structural relief to the high performance anode: external and internal modifications. Specific embodiments of these approaches are shown in Fig. 7. Both approaches provided improvements to the constrictor closure rates. Neither were completely successful, however.

The purpose of the external approach was to provide the required stress relief to alleviate the closure process without affecting the internal flow characteristics. Anode -107 was an attempt to remove the structurally restraining anode outer material and provide the stress relief. The test results were disappointing. The closure characteristic of the -107 anode was not improved over the standard design (Fig. 8). It would appear therefore, that the approach to provide structural relief was not useful. Careful examination of the temperatures, however, revealed that operating temperatures of the -107 anode were approximately 550°C higher than the standard high
performance anode. Increased temperatures were noted above as a major contributor to the closure process. It was proposed therefore, that the stress relief provided by the structural modification was offset by the reduced material strength resulting from the increased anode temperatures.

A second anode was tested that provided stress relief, but yet maintained a similar thermal design as the standard anode. The -111 anode illustrated in Fig. 7, is similar to a standard anode with external relief cuts. The remaining web thickness was approximately 2.0 mm (0.75 inches). The -111 anode was operated for approximately 100 hours. The resulting closure rate shown in Fig. 8, is considerably lower than the standard and thin wall anode designs and therefore suggests an improvement over both. It does not, however, meet the maximum acceptable closure rate. Reduced web dimensions proposed as further improvements were not attempted due to the increased potential for cracking and anode failure.

RRC also investigated an internal approach to provide stress relief in an attempt to reduce the rate of constrictor closure. Figure 7 illustrates the placement of internal relief slots in the constrictor of the -304 anode. These slots are very small and did not provide a complete flow path across the constrictor. The anode was operated for nearly 150 hours.

Initial test results were very promising. The constrictor area actually increased during the first 10 hours of operation. The trend, however, reversed with time and the closure accelerated to a rate similar to the standard anode design (Fig. 8). It is believed that although the constrictor region could not support hoop stress due to the slots, the surrounding material is affected by the same processes and caused the deformation. Attempts to further increase the depth of the slots resulted in adverse stability and performance effects.

Method 3 – Alternate Materials

A final option for resolving the constrictor closure issue was to use an alternate material that had increased strength at the expected operating temperatures. Selection of an advance material was dependent on a number of factors. The first criteria addressed the ability of the material to correct the constrictor closure. The material must have an increase in strength at high temperatures and the high strength must be maintained over the life of the thruster.

The second criteria addressed physical factors that pertain to fabrication and integration into a thruster design. Factors such as machineability, joinability to other materials, compatibility with high temperature hydrazine and its constituents were considered.

The third is availability. A number of high strength, high temperature materials have been developed in laboratories, but are not readily available. Lack of a sufficient quantity of test material prevented their use.

Changing of the anode material provided the greatest improvement to the constrictor closure process. Based on the material investigation, RRC identified a number of material candidates. Availability reduced the list to only two materials: W-2ThO$_2$ and W-4Re-HfC.

A solid W-2ThO$_2$ anode (-108) was fabricated to the same dimensions as the baseline anode. The anode (Fig. 7) was tested under similar operating conditions for approximately 50 hours. The results of the test are shown in Fig. 9. The initial increased strength of the W-2ThO$_2$ material resulted in minimal closure for the first 10 hours. W-2ThO$_2$, as with the tungsten alloy, recrystallizes when maintained at high temperatures for extended periods. Recrystallization reduces the strength of material. The observed increase in the constrictor closure rate after 10 hours is believed to be the result of the reduced material strength properties of the recrystallized W-2ThO$_2$.

Testing of the W-4Re-HfC material in anode -306 was extremely successful. Initial tests indicated that the closure rate was less than the maximum acceptable rate. Based on this success, the -306 anode was placed in an extended duration test.

Extended Duration Test Results

The -306 anode assembly shown in Fig. 7 was tested for 565 hours. The assembly is a W-4Re-HfC anode insert press-fit in a standard tungsten body. The test successfully verified extended thruster life at a mission average performance level of 600 s specific impulse.

Thruster operation was very stable and extended life was expected. At approximately 550 hours, a test anomaly resulted in a very low flow rate. Delivered N$_2$H$_4$ flow was reduced to less than 25.0 mg/s and
resulted in a concentrated arc attachment within the constrictor. Continued thruster operation would have caused additional damage to the constrictor. Further testing was therefore terminated.

**Demonstrated Performance**

The main objective of the test was to investigate the effects of a new material on the rate of constrictor closure. The approach to meet the objective was to proceed slowly with increasing performance and determine a level where the material may lose its strength and begin to close. The resulting performance profile, therefore, began at 550 s Isp and increased in steps to a final performance level of 620 s Isp. The performance change was the result of a controlled reduction in flow rate.

Figure 10 is a graph of the specific impulse data for the life of the test. The performance was continually increased except from 435 to 465 h. The flow rate was purposely adjusted upward during this period to review the closure characteristics before extended operation at 620 s.

Also shown in Fig. 10 is a curve of the real time mission average Isp. The final mission average Isp is shown. Thruster operation for the full performance range proved to be very stable. Operation at higher performance levels was expected to be stable consistent with the performance demonstration of 675 s at 2000 W.

**Life Issues**

**Constrictor Closure Rate**

The results of the extended duration test are very encouraging. Figure 9 is a comparison of the normalized closure rate of the -306 anode and other previously tested anodes. The curve representing the flight-type anode is considered the maximum acceptable closure rate. As is shown in this graph, the -306, W-4Re-HfC anode assembly has demonstrated a closure rate at or below the maximum acceptable rate for 550 hours.

The actual constrictor dimensions were not measured during test. Based on the initial calculations for the parameter \( \frac{P_C}{\sqrt{m}} \), however, the constrictor diameter change was approximated and shown in Fig. 11 Intermediate unpowered tests and visual observations concurred with this trend. Damage resulting from thruster operation with the blocked gas generator did not allow for an accurate post test measurement.

**Cathode Erosion**

Figure 12 is a picture of the cathode tip. The tip profile and the tip characteristics are similar to those of cathodes removed from life tests of lower performance thrusters. Figure 12 illustrates the cathode recession rate of the high performance thruster configuration compared to the erosion rate of the flight type thrusters. The total cathode tip recession of 0.75 mm (0.030 inches) is greater by a factor of 1.5 than the flight type thruster. No adverse effects such as an inability to start, or unstable performance were observed during the extended test.

**Insulator Degradation**

Alumina cathode insulators were continually failing during the early high performance tests. A design modification of the insulator was included in this test. The modification increased the clearance between the alumina insulator and the cathode in the high temperature region.

The design modification proved to be successful. The alumina to cathode fusing process observed on previous thrusters was eliminated. Fracturing of the alumina was also eliminated. Some degradation of the insulator face was observed, however. Excessive heating of the surface was observed. Gas injection across the face from the vortex injector resulted in minor surface erosion. Minimal dimensional modifications are expected to resolve this problem as well.

**Conclusions**

The final goal of the NASA Arjet Thruster Development Program is to demonstrate 600 s NMA Isp at 1800 W for 1000 h. The high performance level was demonstrated early in the program with a thruster configuration based on SOA arcjet designs with a pure tungsten anode. The baseline thruster, however, encountered significant life issues such as start erosion, insulator degradation, cathode erosion and constrictor closure. The life issues were addressed in the development effort.

The most difficult problem to resolve was the constrictor closure issue. The program investigated three different methods to resolve constrictor closure: modified thermal designs, modified
structural designs and alternate materials. The use of alternate materials provided the greatest benefit.

The development effort culminated in a 550 h demonstration of 600 s NMA Isp arcjet. The success of this program provides substantial evidence that an arcjet capable of greater than 600 s will be available for flight qualification and application in the near term.

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References

14. King, D.Q. Arcjet Analysis 12/16/91
Fig. 1. Laboratory Model Thruster

Fig. 2. Power Effects on Specific Impulse

Fig. 3. Power Effects on Efficiency
Fig. 4. Cathode by Profile Comparison

Fig. 5. Cathode Tip Recession

Fig. 6. Baseline Anode Characteristics
Fig. 7. ATD Test Configurations
Fig. 8. Structural Relief Approach

Fig. 9. Anode Material Effects

Fig. 10. Extended Duration Test Performance Results

Fig. 11. Estimated Constrictor Diameter Change

Fig. 12. Post Test Cathode SEM Photograph